Engineering Notes

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Application of Nuclear Solid-Core Rockets to Interplanetary Orbital Launch by Multiorbit Injection

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Nomenclature

a,e	=	semimajor axis and eccentricity, respectively
F_r,F_s	=	radial and horizontal components of acceleration
f	=	true anomaly
I_{sp}	=	specific impulse
n	=	mean daily motion
r,r_p	=	radius and radius of periapsis, respectively
t,t_b	=	time and burn time, respectively
W	=	weight
γ	=	flight-path angle
ΔV	=	change in velocity

Introduction

THE success of the nuclear solid-core rocket program indicates that a nuclear rocket with a 75,000-lb thrust, an 825-sec specific impulse, and a 60-min engine lifetime can be developed by 1975.[1] The use of a 75,000-lb thrust to propel a 1.5- to 2-million-lb spacecraft out of Earth orbit by direct injection results in large gravity losses.[2,3] Consequently, engine clustering, parallel staging, and development of higher-thrust engines received major consideration in studies for orbital launch of large interplanetary spacecraft. Little consideration has been given to changing the injection flight plan to reduce gravity losses. A planetary-mission injection technique that minimizes gravity losses[3] is used herein to evaluate the performance of nuclear solid-core engines for orbital launch of spacecraft that have initial gross weights of 1.5 to 2.25 million lb. (See Fig. 1 and next section).

Multiorbit Injection Technique

The "multiorbit injection" technique (Fig. 1) consists of a series of thrusting subarcs separated by elliptical orbits about the Earth. Each subarc is centered around the periapsis of the elliptical orbit so that all thrusting is accomplished as near the Earth as possible. In the limited case of an infinite number of subarcs, all thrusting is at the periapsis and at right angles to the radius so that gravity losses are zero for the part of the injection between assembly orbit and escape velocity. For that part of the trajectory above escape velocity, only a single thrust arc is possible. Therefore, the effectiveness of the multiorbit injection technique in reducing gravity losses is greatest below escape velocity. However, a carryover effect into the final burn is possible because the final burn is started before periapsis. The magnitude of the gravity losses is a function of the initial thrust-to-weight ratio and the number of orbits until escape.

Steering Equations, Switching Logic, and Assumptions

The thrust steering equations used for this analysis maintain the periapsis of all orbits at a constant value, and the engine switching logic optimizes the lengths of the burns, subject to the constraints that the periapsis altitude is constant and that the period of the final orbit before escape does not exceed 2 days. The constant-periapsis steering requirement is not an optimum steering situation; however, any penalties resulting from nonoptimality of the steering equations are larger for low thrust-to-weight ratios than for higher thrust-to-weight ratios because of the longer thrusting arcs.

The constraint that $dr_p/dt = 0$ is developed into a steering equation by the use of several equations based on a two-body spherical solution to the equations of motion.[4] If $r_p = a(1 - e)$, then

$$dr_p/dt = (1 - e)da/dt - a(de/dt) \tag{1}$$

By setting $dr_p/dt = 0$, and evaluating the derivatives, Eq. (1) can be used to find the ratio

$$F_r/F_s = [(r^2/r_p^2) - 1]/\tan\gamma \tag{2}$$

which is the cotangent of the angle of the desired thrust vector with respect to the radius vector. The technique used for determining engine on and off times is the switching logic described in Ref. 5.

The trajectories used in the analysis are based on two-body orbits about the Earth. The thrust and specific impulse are assumed constant, and the engine startup and shutdown times are assumed to be instantaneous. Likewise, any thrust resulting from cooldown is neglected. However, the amount of propellant required for each cooldown is deducted, because this propellant has an effect on the results. The cooldown propellant weight is $W = 1500 + 2.147\ t_b$. Because of the 1500-lb requirement for zero full-power burn time, the cooldown fuel requirements increase as the number of engine starts increases. As the number of orbits increases, the increase in coolant required resulting from more engine

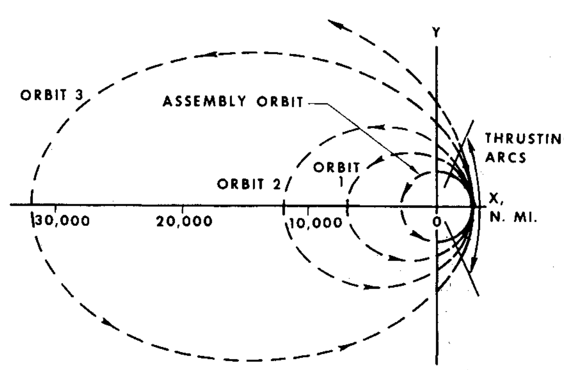

Fig. 1 Multiorbit injection profile for three orbits until escape.

Received May 22, 1969; presented as Paper 69-535 at the AIAA 5th Propulsion Joint Specialist Conference, U.S. Air Force Academy, Colo., June 9–13, 1969; revision received January 16, 1970.

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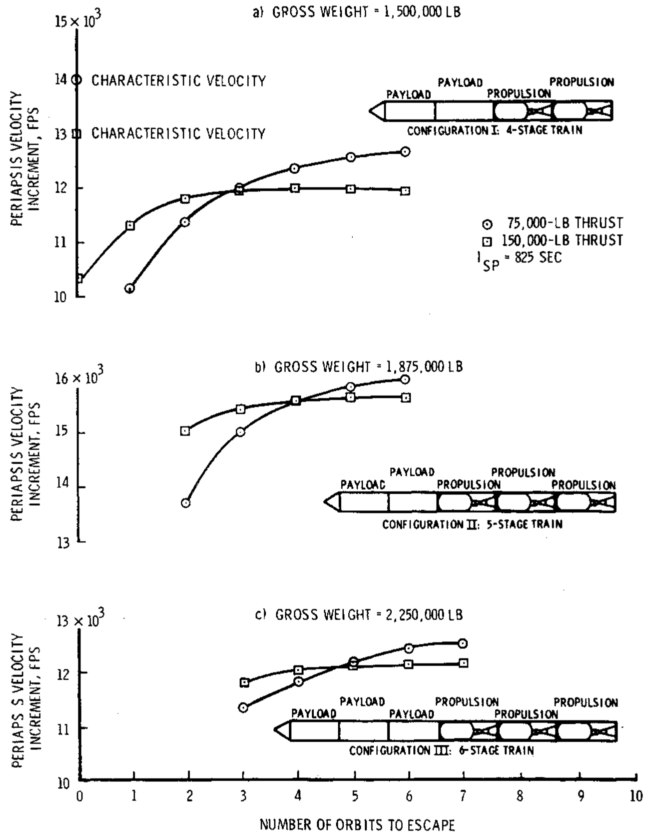

Fig. 2 Performance of vehicles from a 100- by 400-naut mile assembly orbit.

starts becomes larger than the savings resulting from decreased gravity losses.

The spacecraft and propulsion stages are assumed to be assembled in Earth orbit through the use of a series of launches of similar launch boosters. The constraints placed on the propulsion-to-payload ratio are considered in this analysis. A single launch payload is assumed to be either all interplanetary payload or all propulsion. Propulsion stages are only those stages required to inject the payload out of Earth orbit onto interplanetary mission orbit. The interplanetary propulsion requirements are considered to be payload. The spacecraft, called a stage train, consists of a series of equal-weight modules assembled in orbit.

Four-, five-, and six-module stage-train configurations were chosen for analysis (Fig. 2). The assembly, or initial, orbit is assumed to be elliptical (100 by 400 naut miles). Each stage has a gross weight of 375,000 lb in the assembly orbit. Thus, the gross weights of configurations I, II, and III are 1.5, 1.875, and 2.25 million lb, respectively. The propulsion-stage weights are calculated based on a mass fraction of 0.20 of the fuel weight for the tanks and on 0.275 of the thrust for the engine, thrust-structure, and interstage.

Results

The performance of each configuration is calculated for 75,000- and 150,000-lb thrusts, representing a single engine and a pair of engines, respectively. The performance is presented as the periapsis velocity increment required by the injection maneuver. This increment is comparable directly with the velocity increment required for an impulsive maneuver. The multiorbit injection technique greatly improves the performance of all configurations. The losses from direct injection are so high for most configurations that they are not included in the figure. The best performance for direct injection is obtained with 150,000 lb of thrust in configuration I (Fig. 2a). The losses were approximately 1600 fps or 13% of the total velocity increment for the three-orbit injection maneuver with the same configuration. The improvement for the 75,000-lb thrust level vehicle of configuration I is even greater than for the 150,000-lb thrust level vehicle. In Fig. 2a, the performance of the 75,000-lb thrust level vehicle for five and six orbits approaches the characteristic velocity of the 150,000-lb thrust level vehicle. The multiorbit injection maneuver, therefore, makes practical the consideration of thrust-to-weight ratios of 0.05, represented by 75,000 lb of thrust pushing a 1.5-million-lb spacecraft, because this thrust-to-weight ratio outperforms the 0.10 thrust-to-weight ratio represented by the 150,000-lb thrust. A thrust-to-weight ratio of 0.0333, represented by a 75,000-lb thrust on configuration III, does not seem unreasonable. Subject to the assumptions of cooldown propellant loss and instant startup, the 75,000-lb thrust outperforms the 150,000-lb thrust if no restriction is placed on the number of orbits. Higher thrust levels are required only if a restriction is placed on the maximum number of injection orbits.

The most likely restrictions are the injection time and the radiation-shielding requirements. The injection time required for the seven-orbit escape maneuver is approximately 3 days. This is insignificant when compared with total mission times of 300 to 1000 days for interplanetary round trips. The preliminary requirements to shield against solar-flare radiation for interplanetary missions are specified as 10 g of aluminum for each square centimeter of shield. For multiple passes through the Earth radiation belts, no more than 5 g of aluminum for each square centimeter of shield is required.

The increased performance of the 75,000-lb thrust over the 150,000-lb thrust is primarily attributed to the 1000-fps reduction in characteristic velocity resulting from the increased weight for the higher thrust levels and from smaller cooldown propellant losses. Because some thrust occurs during cooldown, it has been suggested that a gain in velocity could be obtained. However, any gain in velocity from the thrust of approximately 3000 lb during cooldown is expected to be small because 1) although a specific impulse as high as 400 sec may be possible, a 60% loss still occurs; 2) a reduction of 25 to 1 in thrust level will cause the gravity losses during cooldown to be considerably greater than during the main thrust phase; and 3) cooldown occurs in trajectory regions which have higher gravity losses than the main thrust phase. Therefore, it appears justifiable in this analysis to assume that the propellant required for cooldown is a complete loss.

A performance comparison of the three configurations and the requirements for interplanetary missions can be made from Fig. 3.[6] The two dashed lines represent the injection velocity increments for the three configurations with a 75,000-lb thrust level. Configurations I and III can inject payload stages on 25 of the 34 missions. Six of the nine

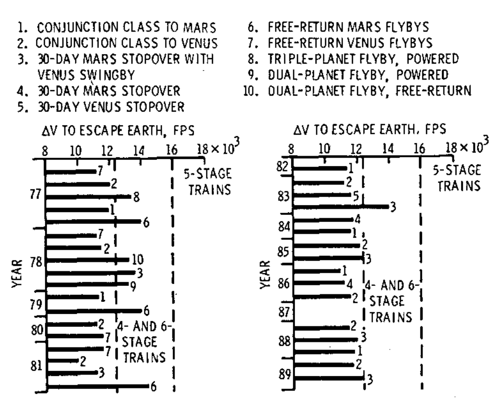

Fig. 3 Orbital-launch velocity requirements for 34 interplanetary missions (escape from a 100- by 400-naut-mile orbit).

missions requiring higher injection velocities are flyby missions which should not require two- and three-payload stages. The 1985, 30-day Mars-stopover mission with an outbound Venus swingby requires only 200 fps more ΔV than is available. Any performance gains resulting from engine performance improvements will place this mission within the capabilities of the discussed configurations. Only the 1978 Mars-stopover mission with an outbound Venus swingby and the 1983 Mars-stopover mission with an inbound Venus swingby require configuration II.

Conclusion

The results of this analysis have a significant impact on the necessary systems and technology developments for manned interplanetary missions. Studies based on direct injection techniques[2] indicate that thrust levels higher than 75,000 lb would result in payload gains of 9 to 16% for manned interplanetary missions. These thrust levels could only be obtained by a new engine, parallel engines, or parallel staging developments, used separately or in combination. These developments would require longer lead times and have higher costs than the 75,000-lb-thrust engine development. Any one of these operationally complex developments may require an expensive test program for development. As shown by this analysis, a performance penalty rather than a performance gain may result. The multiorbit injection technique eliminates the need for these developments.

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Real-Time Compression and Transmission of Apollo Telemetry Data

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FLIGHT control for NASA manned missions is conducted at the Mission Control Center (MCC) in Houston. Telemetry data from up to 800 sensors in the spacecraft are relayed from the remote tracking sites via the worldwide NASA Communications Network (NASCOM) based at Goddard Space Flight Center to the MCC. Two 2400 bps digital high-speed data (HSD) circuits and one voice bandwidth analog circuit from each site to Houston are provided by the NASCOM network. Up to four telemetry downlinks with bit rates of 51.2 kbps or 72 kbps each are received at the remote site. Presently, by a process of repetitive sample selection, processors at the remote site reformat the data to fit the capacity of the NASCOM circuits. Data compression is an alternative processing technique which will most efficiently use transmission bandwidth and is being considered for the future expansion of telemetry requirements.

Limited results from early Apollo flights indicate that the telemetry data appear to be sufficiently redundant that a data-compression scheme can be used.

The compression technique discussed in this Note is the zero-order predictor whose operation is described as follows. For each sensor output, a tolerance corridor of $\pm K$ (expressed in percent of full-scale range) is established around the first sample y_0 which automatically is considered significant. The next sample y_1 is read in and compared with y_0 as $|y_1 - y_0| < K$. If the inequality is satisfied, y_1 is discarded as redundant and the next sample y_2 is compared with y_0 in the same way. If the inequality is not satisfied, y_1 is output as a significant sample and a new corridor is established around y_1. Then y_2 is compared for $|y_2 - y_1| < K$ and the process continues in like manner. The maximum peak error is $\pm K$. Reconstruction is effected by drawing a straight line of zero slope beginning at each significant sample and continuing until the time of the next significant sample.

The present operational system is described because it provides a standard of comparison for a system using data compression. Problems arising from highly active data and false activity are examined, important considerations of system implementation are identified, and proposed design approaches are described.

Present System

The data flow through the system is as follows. Four PCM telemetry links carry the telemetry data from the Apollo vehicles once they reach Earth orbit. The links are from the Lunar Module (LM), Command and Service Modules (CSM), the Saturn booster third stage (S-IVB) and its Instrumentation Unit (IU). The links are received at whichever remote site is in line-of-sight of the vehicles. The data are decommutated and processed for transmission over the high-speed data circuits of the NASCOM network. Data compression will take place in the Remote Site Data Processor (RSDP) by a modified UNIVAC 642B computer with 65,000 words of core. Reconstruction will take place at Mission Control Houston by the Command Communications and Telemetry System (CCATS) and the Real-Time Computer Complex (RTCC). CCATS has a complex of three UNIVAC 494s, each with 131,000 core and drum storage of 1,310,000 words. The RTCC uses IBM 360-75 computers with 370,000 core and 1,048,000 words of auxiliary storage.

The types of telemetry data are 1) sampled 8-bit and 10-bit coded analog waveforms, 2) events or bilevels, 3) 16- and 26-bit data words from onboard computers (CMC, PGNS, LVDC), and 4) special words. The sampling rates range from 0.5 to 200 samples per second. During Apollo flights, the MCC monitors a total of approximately 1100 of the measurements. All of the measurements represent a remote site input-data rate of approximately 100 kbps. From the remote site to the MCC, a total high-speed line capacity of 4.8 kbps is available. In order to accommodate the wide differences in data rates between the space-to-ground and NASCOM links, two primary methods of data-rate reduction are used. The first method is sample rate reduction. For example, a given

Received August 28, 1969; revision received February 18, 1970. The UNIVAC contribution to this Note was performed in conjunction with services provided for NASA Goddard Contract NAS 5-10006. MITRE Research sponsored by the NASA Manned Spacecraft Center under Contract F19(628)-68-C-0365 with Electronic Systems Division, Air Force Systems Command.
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